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Effect of loading spectrum clipping and truncation on fatigue crack growth behavior of 7475-T7351 aluminum alloy under variable amplitude loading

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Abstract

The paper summarizes conclusions based on fatigue crack growth experiments and numerical calculations focused on full commuter aircraft spectrum optimization. The fatigue crack growth experiments were performed on 7475-T7351 alloy. Sequences with different truncation (omission) and clipping stress levels were investigated. All experimental data are supported by extensive numerical crack growth analyses using FASTRAN and AFGROW software. The sequence with a clipping of 2.78 g values and omission level of 10 MPa was assessed as an optimum from point of view of time saving and test credibility. Omission of any higher stress range affects the fatigue crack propagation significantly.

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Keywords: variable amplitude loading; truncation; clipping; 7475 aluminium alloy; fatigue

1. Introduction

Test sequence definition is one of basic inputs into the damage tolerance evaluation and lifetime structure certification. The full spectrum of any type of aircraft gives non acceptable length of loading sequence from the view point of experimental proof conducting mainly with respect to full-scale structures. Therefore, generated or in-service

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measured stress sequences need to be shortened with the aim of time and cost savings. Simultaneously, the shortened load sequence should not significantly influence behavior of the structure during the proof in an undefined or unknown way.

There are two basic ways to reduce the length of a random flight-by-flight sequence:

- Clipping of rarely occurring high loads;
- Truncation (omitting) of numerous very small load cycles.

The effect of clipping of rarely occurring high loads is well known [1]. It results in shorter crack growth lives for lower clipping levels, which is the reason why the high load levels should be avoided in full-scale flight-by-flight simulation tests in order to obtain conservative results. It is good practice (suggested by Schijve in [2]) to use a clipping level of such a size for loads that occur less frequently than about 10 times in an aircraft life time (a low occurrence probability).

In the case of small amplitude load cycles, these do not contribute considerably to crack growth, especially in view of the retardation effect. Truncation (omitting) of small amplitude cycles could result in a significant acceleration of experiments and an appreciable shortening of testing time. Nevertheless, sometime the truncation leads to a longer crack growth life in flights. Hence, a truncation level for small amplitudes omitting should be specified very carefully and on the basis of experiments or simulations.

Nomenclature

a	average crack length measured from specimen centre line (in mm)
FH	flight hours
L	longitudinal direction
M(T)	middle-crack-tension specimen
T	transverse direction
v	fatigue crack growth rate (in m/cycle)
VZLU	Aeronautical Research and Test Establishment
α_1, α_2	stress constraint constants
ΔK_{eff}	effective value of stress intensity factor range
σ_{1g}	mean stress value corresponding to a load of 1 g (in MPa)
σ_{max}	maximum stress in a load cycle (in MPa)
σ_{min}	minimum stress in a load cycle (in MPa)

The paper summarizes conclusions based on fatigue crack growth experiments and numerical calculations focused on the full commuter aircraft spectrum optimization from point of view of the sequence shortening. The full commuter aircraft spectrum was considered as a base for comparison and evaluation. The fatigue crack growth experiments were performed on 7475-T7351 aluminum alloy usually used for integrally stiffened wing panels.

Several modified sequences were tested, calculated and compared to the full sequence. Sequences with different truncation (omission) and clipping load levels were investigated. The full sequence was optimized with regard to a limitation of its modification effect on the fatigue crack propagation. Moreover, in the case of the clipped sequence, the sequences generated before and after clipping procedure of the data matrix were compared for the sake of evaluation of the generation and clipping interaction. All experimental data are supported by numerical crack growth analyses using FASTRAN model and AFGROW software.

2. Experimental methods

2.1. Specification of sequences

A full loading sequence typical for a commuter aircraft was derived from data obtained during a full-scale wing test. The full sequence contains 1 684 980 loading peaks and simulates 2768 flights (3000 flight hours) typical for a stringer site between two ribs of the commuter aircraft bottom wing panel. The maximum load factor in the full load sequence was 3.24 g and the minimum values were limited at a range level of 0.20 g. The blocked load-time full sequence was generated in stress values (MPa) representing a loading of the structure critical part using CESAR software [3]. This software was developed by VZLU in frame of FP7 EU project CESAR.

Individual flight phase's spectra were simulated using 14 load factor levels and the flights were separated by a decrease in load from a mean stress level in corresponding flight down to negative values. These values represented a taxiing of the commuter aircraft on the ground. The low-high and high-low peaks ordering are used. A graphical visualization of a part of generated full reference loading sequence is shown in Fig. 1, where the part designated by a rectangle shows the section with maximum load peaks occurrences in the sequence (Fig. 2). Maximum stress value occurring in the full sequence was $\sigma_{max} = 111.4$ MPa and the minimum stress value was $\sigma_{min} = -33.4$ MPa.

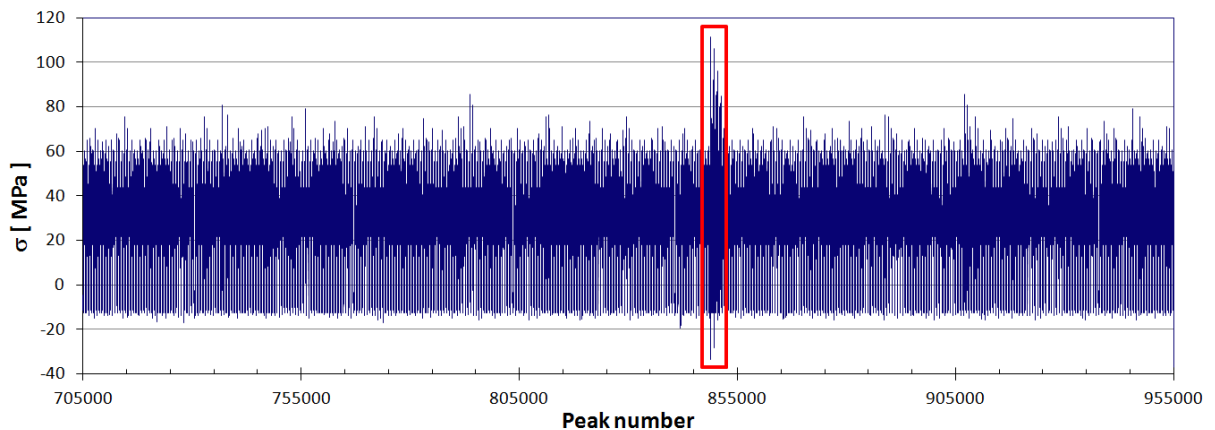


Fig. 1. Example of a part of the generated full loading sequence.

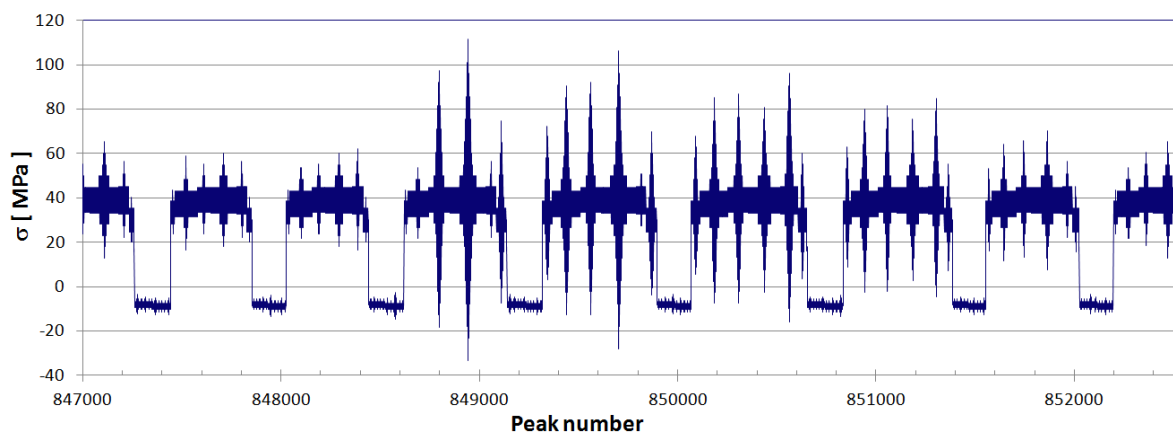


Fig. 2. Detail of section with maximum load peaks occurrences in the generated full sequence (designated by the rectangle in Fig. 1).

Following different types of sequences derived from the full sequence were prepared and experimentally tested:

- Reference sequence – the complete (full) sequence;
- Seq. A – the full sequence with a clipping level of 2.78 g;
- Seq. B – a sequence generated after clipping of input data (level of 2.78 g);
- Seq. C – a truncated sequence (lower amplitudes than 10 MPa omitted) and with a clipping level of 2.78 g;
- Seq. D – a truncated sequence (lower amplitudes than 15 MPa omitted).

Parameters (such as the omission and clipping stress level, the sequence length, etc.) characterizing individual sequences are summarized in Table 1.

Table 1. Different types of commuter aircraft load sequences.

Sequence type	Omission stress level	Clipping level / Max. stress σ_{max}	Min. stress σ_{min}	Sequence length (peaks)	Number of cycles/flight	Percentage change of number of cycles
Ref. sequence	–	– / 111.4 MPa	-33.4 MPa	1 684 980	304	–
Sequence type A	–	2.78g / 97.4 MPa	-33.4 MPa	1 684 980	304	0.0 %
Sequence type B	–	2.78g / 97.9 MPa	-20.6 MPa	1 685 104	304	0.0 %
Sequence type C	10 MPa	2.78g / 111.4 MPa	-33.4 MPa	1 168 640	211	-30.6 %
Sequence type D	15 MPa	– / 111.4 MPa	-33.4 MPa	284 182	51	-83.1 %

Furthermore, an effect of truncation level on fatigue crack growth life for the full reference sequence was assessed using FASTRAN model calculations. Summary of the various omitting stress range levels is shown in Table 2.

Table 2. Summary of the various truncation stress range levels for FASTRAN model calculations.

Sequence type	Omission stress level	Sequence length (peaks)	Number of cycles/flight	Percentage change of number of cycles
Ref. sequence	–	1 684 980	304	–
Truncated seq. (C)	10 MPa	1 168 640	211	-30.6 %
Truncated sequence	11.5 MPa	1 064 330	192	-36.8 %
Truncated seq. (D)	15 MPa	284 182	51	-83.1 %
Truncated sequence	20 MPa	250 124	45	-85.2 %

2.2. Material

Samples taken from AW7475–T7351 (Al Zn5.5MgCu) aluminum alloy plate (Table 3) of 76 mm in thickness, made by ALCOA manufacturer, were used for the experiments. The delivery condition and heat treatment designated as T7351 for aluminum tempering consists of solution annealing (usually at 465–475 °C), water quenching (at 25–40 °C), controlled stretching (1.5–3 %), and two-step artificial aging (usually at 105–125 °C for 4–24 hours and 165–180 °C for 5–20 hours). Corresponding average tensile properties of this semi-product are 419/486 MPa and 15 % for the proof strength/ultimate tensile strength and percentage elongation, respectively [4].

Table 3. Composition of the experimental material (in wt.%).

Element	Si	Fe	Cu	Mn	Mg	Cr	Ni	Zn	Ti	Zr	Al
Content	0.037	0.039	1.537	0.007	2.164	0.202	0.002	5.787	0.040	0.003	Balance

2.3. Fatigue crack growth testing

Samples of dimensions of 250×100×8 mm were cut from the plate by means of electro-discharge machining for the fatigue crack growth rate tests in L-T direction. Five flat test specimens of middle-tension, M(T), type with a central double-sided notch were manufactured according to ASTM E647 standard [5]. The test specimen geometry is shown in Fig. 3. Prior to a hole and notch manufacturing, the surface layer was milled off in the middle of both sides of the specimen in order to obtain a thickness of 3.6 mm for the sake of a real aircraft part simulation. Afterwards, a hole of 2 mm in diameter was drilled concentrically and a double-sided notch (5 mm in length) was machined symmetrically using spark erosion method in each test specimen.

The fatigue crack growth rate tests were performed using a servohydraulic Hydropuls Schenck load frame (load capacity of ±250 kN) controlled by Instron 8800 electronic unit and the FastTrack 2 software. The test specimens were gripped in their entire width using standard mechanical-wedge clamping devices during testing. Fatigue crack propagation was monitored visually using marks with a spacing of 1 mm by means of Olympus SZ40 light microscopes (up to 40× magnification). The crack growth monitoring was carried out on both sides and both surfaces of test specimens, i.e., four cracks were observed for each specimen during testing.

In the first fatigue precracking stage, an initial fatigue crack of a minimum length of 4 mm was prepared symmetrically on both sides of the test specimen using cyclic loading with a sinusoidal form and constant amplitude ($\sigma_{max} = 55$ MPa and stress ratio $R = 0.1$). The second stage of testing, the variable amplitude loading, was performed using the sequences documented in Table 1 at a loading frequency $f = 7$ Hz and at room temperature in laboratory environment.

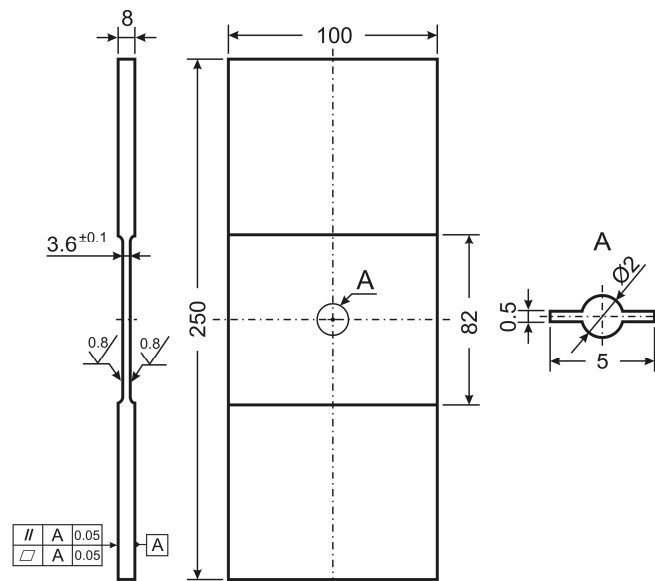


Fig. 3. Test specimen geometry.

2.4. Data evaluation

The aim of experimental work was to define the crack length versus number of cycles or flight hours (FH) or flights dependence. Data evaluation was performed for each test specimen in compliance with ASTM E647 [5] standard and MMPDS database [6] requirements.

Experimental values of four surface fatigue crack lengths referenced to the longitudinal axis of symmetry (measured on both sides and both surfaces of each specimen) were averaged using simple interpolation method for the each measured crack length. No significant asymmetry was observed during experiments performed.

2.5. Numerical calculations

Calculations and simulations of the fatigue crack growth under variable amplitude loading were performed using the FASTRAN retardation model [7], version 3.82, implemented in the AFGROW computer software [8], version 5.01.01.16. The model is based on the Dugdale's strip-yield model [9], but modified to leave plastically deformed material in the wake of the crack. The most important features of the FASTRAN model is its ability to model three-dimensional constraint effects and its possibility to objectively measure input parameters using standard M(T) specimens, including parameters for an initial plane strain state and its transition into plane stress state [10].

Material characteristics were measured at a constant amplitude loading [4] according to ASTM E647 standard [5] and approximated by the Paris' law in the form of $\nu = 1.92321 \cdot 10^{-10} \cdot (\Delta K_{eff})^{3.134}$. The FASTRAN model equations (ΔK_{eff} decomposition algorithm) were used for the assessment of the effective value of the stress intensity factor. The stress constraint constants were evaluated as $\alpha_1 = 2.679$ and $\alpha_2 = 1.00$ according to VZLÚ methodology [10].

3. Results

Based on the data referred in Tab. 1 and Tab. 2, it is evident that the commuter aircraft load sequence could be divided into three groups from the experimental time saving point of view:

- Full sequence together with the clipped sequences A and B;
- Truncated sequences with omitted all stress ranges smaller than 11.5 MPa (sequence C);
- Truncated sequences with omitted all stress ranges between 15 and 20 MPa (sequence D).

The loading peaks number was significantly different for these three groups. The omitting of all stress ranges smaller than 10 and 11.5 MPa induced the sequence shortening about 30 and 37%, respectively, as compared with the full reference sequence. The third group, where omitting stress range level was 15 and 20 MPa, induced significantly higher sequence shortening (in the range from 83% to 85%) as compared to the full sequence. From another point of view, it meant a decrease from given 304 cycles per flight for the full sequence down to 200 and 50 cycles per flight for the lower and higher truncation levels, respectively.

Results of variable amplitude loading experiments using different types of sequence defined in Tab. 1 are presented in Fig. 4a. Corresponding simulation results are shown in Fig. 4b. Both experimental and simulation crack propagation curves were unified to one initial crack length of 7.8 mm for the sake of a comparison simplification. Examples of comparison of the experimental data with the calculated crack propagation curves using the clipped (seq. A and B in Tab. 1) and the truncated (seq. D) sequences in non-unified form (with original initial crack lengths) are presented in Fig. 5.

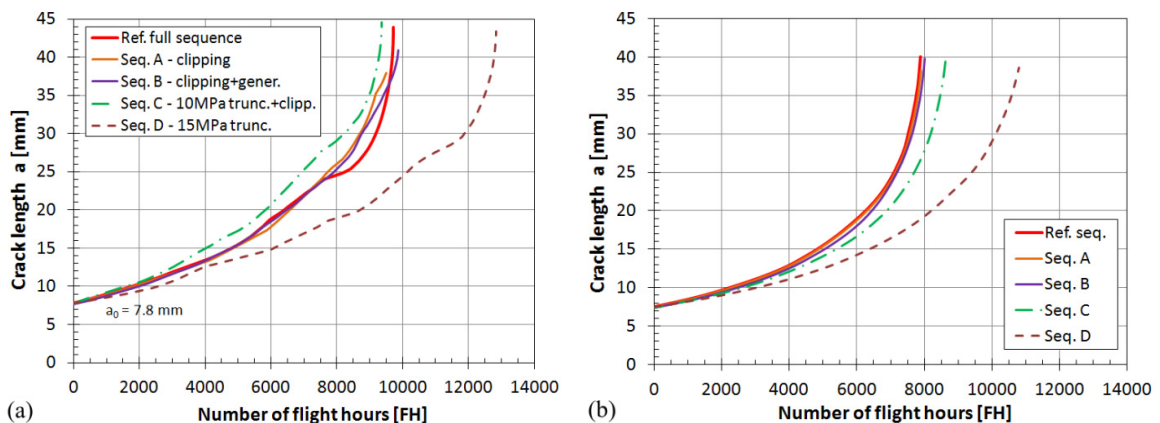


Fig. 4. Graphical presentation of the results of (a) different sequences experiments and (b) their simulations using the FASTRAN model.

The fatigue crack growth experiments under variable amplitude loading showed that clipping level of 2.78 g (from 3.24 g level) exhibited no effect on the fatigue crack growth life time. Moreover, the clipping resulted in disappearance of a significant retardation effect observed at the crack length of about 25 mm in the case of the full reference sequence (Fig. 4a). Hence, a different retardation effects could be observed in corresponding sections of the full and clipped sequences. These effects were not evident in the FASTRAN model calculation curves (Fig. 4b).

Both experimental and calculation results showed that the life time of the sample were not affected by insertion of generation of the sequence before or after the clipping procedure (Fig. 5). From the number of flight hours point of view, the FASTRAN model calculation had a tendency to overestimate slightly the fatigue crack growth data (in case of sequences A and B) as compared to the experimental one. Nevertheless, a total fatigue life time calculated corresponded well to experimentally measured values for the clipped sequences. On the contrary, in the case of truncated sequences, the calculation model behaved conservatively, as the life times were underestimated by about 15 percent (Fig. 5).

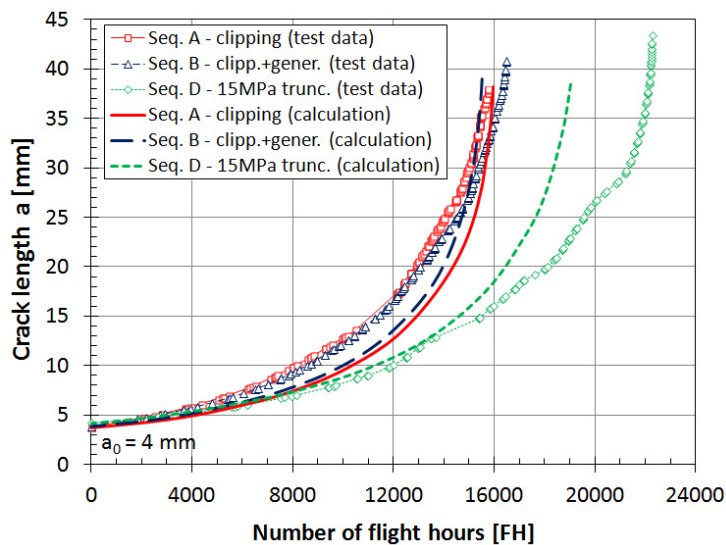


Fig. 5. Crack length against flight hours for experimental data and FASTRAN model calculations in the case of clipping and truncation.

Results of assessment of the truncation level effect on fatigue crack growth for the full reference sequence using FASTRAN model calculations are graphically shown in Fig. 6. According expectations, the higher level of truncation (higher stress range cycles omission) leads to higher number of flight hours. As discussed above, from the experimental time saving point of view, the truncation effect on the full load sequence could be divided into two groups. The first one with omission of 10 and 11.5 MPa caused the flight hours increase of about 9 and 13%, respectively, as compared with the full reference sequence. The second group with omission of 15 and 20 MPa induced significantly higher increase in the flight hours by 37 and 46%, respectively. Hence, it follows that a reduction from given 304 cycles per flight (the full sequence) to 50 cycles per flight for the higher truncation levels resulted in a significantly longer crack growth life in flights that should be unsuitable. Therefore, the sequence C, with clipping of 2.78 g values and truncation of all stress ranges lower than 10 to 11.5 MPa, was assessed as an optimum from point of view of time saving and test credibility in comparison with the full load sequence.

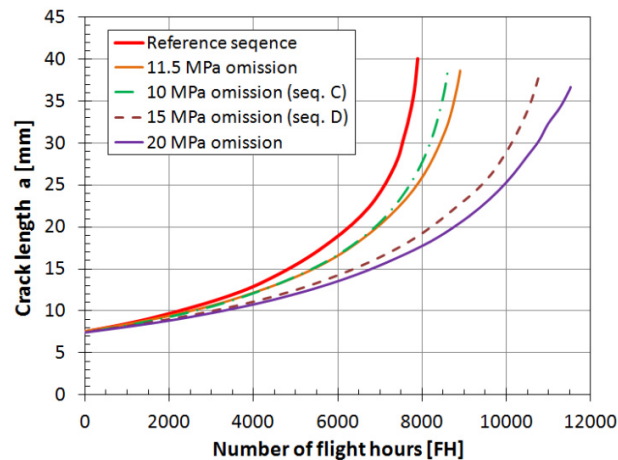


Fig. 6. Comparison of truncation level effect on fatigue crack growth for the full reference sequence (FASTRAN model).

4. Conclusion

The paper presents the study of stress level clipping and truncation effects on the crack propagation behavior of AW7475–T7351 aluminum alloy plate material. The full load spectrum, typical for a stringer site between two ribs of the commuter aircraft bottom wing panel, was taken into account as the base input for the comparison. Different stress levels for omission (truncation) and clipping of 2.78 g is numerically evaluated. At the same time the extensive experimental work was carried out. Based on the numerical and experimental data, the resulting sequence with clipping of 2.78 g values and truncation of all stress ranges lower than 10 to 11.5 MPa was assessed as an optimum from point of view of time saving and test credibility in comparison with the full load sequence (about 9 and 13% flight hours increase). Omission of any higher stress range affects the fatigue crack propagation phase significantly.

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